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Future Reusable Launcher Options for Europe: the Heavy Class Segment

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The heavy class segment has gained new interest also in Europe because of the SpaceX Starship&SuperHeavy launcher configuration under development. In a first step, the paper provides a thorough technical analysis of Starship's estimated capabilities in its early operational phase, based on independent modeling with openly available data.

The main part of the paper is dedicated to the technical evaluation of European options in serving a roughly similar payload class approaching 50 Mg in single launch to LEO. A launcher system analysis looks into Ariane 6 evolution options and explores the technical limits based on the assumption of expendable stages. A significantly better performance perspective can be achieved through a completely new architectures with first stages designed reusable and exclusively liquid cryogenic propellants chosen. Both, VTVL and VTHL designs are compared including different return technologies with related strong impact on system performance. Fully reusable, vertically landing configurations are addressed complementary to the small ESA-funded PROTEIN-study.

The paper concludes with a comparative evaluation of main technical characteristics of launch vehicle options and indication of promising development roadmaps.

Abbreviations

AOA	Angle of Attack
CAD	Computer Aided Design
DOF	Degree of Freedom
EHLL	European Heavy Lift Launcher
GLOW	Gross Lift-Off Mass
HLS	Human Landing System
IFT	Integrated Flight Test
LCH4	Liquid Methane
LEO	Low Earth Orbit
LH2	Liquid Hydrogen
LOX	Liquid Oxygen
MECO	Main Engine Cut Off
MR	mixture ratio
OTV	Orbital Transfer Vehicle
RCS	Reaction Control System
RLV	Reusable Launch Vehicle
RTLS	Return To Launch Site
SLB	SpaceLiner Booster stage
SLME	SpaceLiner Main Engine
SLO	SpaceLiner Orbiter stage
SLP	SpaceLiner Passenger stage
TPS	Thermal Protection System
TRL	Technology Readiness Level
TSTO	Two-Stage-To-Orbit
VTHL	Vertical Take-off Horizontal Landing
VTVL	Vertical Take-off Vertical Landing

1 Introduction

Recently, launcher technology and development has become highly dynamic again and the worldwide yearly number of launches has climbed to record-levels, hardly imaginable only few years ago. Key-actors are the US-company SpaceX with its Falcon 9 and several providers in China.

The global competitive landscape could soon be revolutionized with the SpaceX Starship&SuperHeavy launcher configuration when becoming operationally available. More heavy class activities are ongoing: after ULA Vulcan's entry in 2024, in January 2025 Blue Origin's New Glenn achieved its maiden flight and could become another serious player in the field of heavy launch missions. Both aim for reusability of their first stages or propulsion bay in the medium term.

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In this global environment, defining the best European responses to this challenge is becoming increasingly important. ESA initiated the studies NESTS and PROTEIN investigating reusability up to heavy- and super-heavy-lift capabilities. CNES announced the new demonstrator project DEMESURE (DEMonstration Étage SUpérieur REutilisable) in support of technologies for a future reusable TSTO upper stage capable of delivering 20 t to LEO [1].

The main part of the paper is dedicated to the technical evaluation of European options in serving a roughly similar payload class. Mission requirements demand at least the Ariane 6 payload mass to LEO and the goal extends up to 50 tons to LEO or beyond in single launch. Typical future applications which would require a significant number of missions per year are the deployment of mega-constellations, space solar power or deep-space human and large-scale robotic exploration.

1.1 Understanding SpaceX' Starship Launcher

Elon Musk's bold vision of transporting humans in significant numbers to Mars [2] is driving the development of a space transportation system to dimensions and capabilities never seen before. The concept has perceived several technical evolutions in the last 10 years [3, 4]. The SpaceX Starship-SuperHeavy is now flying in prototype configuration and is continuously progressing in mastering its ascent and atmospheric reentry mission. Successful capturing of the SuperHeavy first stage could be achieved multiple times after performing a complex RTLS-maneuver. The fully reusable system is still not yet operational and its present payload capability is limited, not being in focus of the current development process.

Once becoming operational, the two-stage Starship launcher could bring a fundamental shift to global space transportation. The circumstances need to be well understood and therefore require, as a first step, a thorough technical analysis of Starship's actual capabilities in its early operational phase, based on the openly available data. Note, these analyses have been limited to the LEO ascent and return capabilities of Starship-SuperHeavy. SpaceX has been contracted by NASA to develop and operate a Starship derivative as Human Landing System (HLS) on the Moon within the Artemis-program [5], requiring additional features (like tankers for on-orbit refilling) which are not analyzed.

1.1.1 Architecture

The SpaceX Starship-SuperHeavy is designed to become the first fully-reusable space transportation system to LEO. Defined as TSTO with both stages in tandem arrangement, the vehicle is using the propellant combination LOX-LCH4. Analogous to the operational Falcon9, the engines of the main propulsion system are similar on both stages with main difference in nozzle expansion ratio. The engines have been named Raptor and are using the hydrocarbon propellant methane, which one day should be produced in-situ on Mars [2]. The medium specific performance of LOX-LCH4 is partially offset by selecting for Raptor the closed cycle at extremely high operating conditions.

The primary purpose of the Starship assembly is to support deep-space human exploration and colonization and thus requires a payload capacity beyond 100 tons in LEO. GLOW is above 5000 tons, roughly twice that of the Saturn V in the Apollo-moon program. The total length of the launcher exceeds 120 m with a fuselage and tank diameter of 9 m. A more detailed technical description of SpaceX' Starship-SuperHeavy and how the launcher has been remodeled by DLR is described in reference 6.

1.1.2 Raptor main propulsion system

The SpaceX Raptor operates in Full-Flow Staged-Combustion (FFSC) cycle which is using the complete propellants to drive the turbopumps. Raptor is the first FFSC rocket engine ever flown while the RD-270 (8D420) was already in the Soviet Union in the late 1960s the first ever designed and tested FFSC-engine by Energomash.

Precise engine performance data of the Raptor 2 has not been published. Therefore, an independent DLR-analysis of Raptor 2 has been performed using the rocket cycle calculation tool RPA. The SpaceX announcement of 230 tons (2260 kN) of sea-level thrust [7], the chamber pressure of 30 MPa with assumed mixture ratio of 3.6 and nozzle exit diameter 1.3 m serve as guidelines for the calculation.

The Raptor 2 data based on DLR-calculations while at the same time in overall agreement with references [7, 8] are a good working baseline for launch vehicle analyses. Shortly after lift-off the SuperHeavy&Starship-launcher significantly throttles back to reduce loads in the maximum dynamic pressure regime. This change in massflow might be achieved by adapting mixture ratio (MR), chamber pressure or both. Conditions for the engine with same thrustchamber geometry working at 250 bars are listed in the right column of Table 1 showing a reduction in sea-level thrust of almost 18%. The sensitivity of MR-variation on the Raptor 2 performance is depicted in [6].

Last year SpaceX announced the production start of its latest variant Raptor 3 [9] with capability of operating at very high chamber pressure of 350 bars. This engine should incorporate highly innovative features, like all components regeneratively cooled and at the same time simplifications and significantly increased T/W-ratio. Although,

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functionality of the features is not yet exactly understood, nevertheless, estimation of the engine overall performance is possible assuming same thrustchamber geometries as for Raptor 2 and nominal chamber pressure of 35 MPa.

Calculated thrust levels of Raptor 3 are increasing by 16.6% for the sea-level variant (Table 2) and slightly less by 15.6% for the upper stage engine. Any further increase is not compatible with the geometry constraints and announced chamber pressure. Therefore, the posted thrust [9] of 280 t_f (2746.8 kN) is remarkably 4.98% above the calculated thrust at MR= 3.8. Further approaching stoichiometric conditions does no longer surge thrust as the increase from 3.6 to 3.8 is already pretty small. The photograph provided in [9] showing all Raptor variants 1, 2 and 3 is not indicating changes in the thrustchamber geometry. In any case, a surged-up size of the chamber would be hardly compatible with an integration of 33 Raptors in the SuperHeavy base area. However, it is interesting to note that the Raptor 3 operating at 35 MPa with its nozzle assumed at expansion 32 would already be slightly underexpanding at sea-level. This situation is not really advantageous from the performance perspective but is dictated by severe geometry constraints of the SuperHeavy first stage.

Table 1: SpaceX **Raptor 2** engine (sea level variant) in DLR-calculated technical data

Mixture ratio [-]	3.6	
assumed nozzle area ratio [-]	32	
Chamber pressure [MPa]	30	25
Mass flow engine [kg/s]	663.4	553.9
Thrust at sea level engine [kN]	2123	1747
Thrust in vacuum engine [kN]	2260	1884
Specific impulse at sea level [s]	326.4	321.6
Specific impulse in vacuum [s]	347.4	346.8

Table 2: SpaceX **Raptor 3** engine (sea level variant) in DLR-calculated technical data

Mixture ratio [-]	3.6	3.8
assumed nozzle area ratio [-]	32	
Chamber pressure [MPa]	35	
Mass flow engine [kg/s]	765.8	774.9
Thrust at sea level engine [kN]	2476	2479
Thrust in vacuum engine [kN]	2613	2617
Specific impulse at sea level [s]	329.6	326.3
Specific impulse in vacuum [s]	347.9	344.3

A potential redesign option of the SuperHeavy V2 could be the introduction of a larger nozzle for Raptor 3 which would have a better mission adaptation and thus might reach the claimed vacuum thrust level. However, this requires major modifications on the SuperHeavy's aft bulkhead and engine mounting points and with the outer Raptors' ring diameter increased an additional skirt for protection of the engines and is speculative.

The corresponding engine of Raptor 2 for the upper stage with increased nozzle is called *RVAC* by SpaceX. This engine should have an exit diameter of 2.3 m [8]. Thus, nozzle geometry has been used again as baseline assumption in combination with the reasonable hypothesis that other key-parts of the engine (e.g. turbopumps, injector, throat) are similar to the sea-level variant. An expansion ratio of around 100 is consistent with the stated exit diameter.

Table 3 lists key performance data of RVAC calculated under above constraints. According to SpaceX's website, the thrust should reach 258 t_f (2530 kN) [8], at least approximately 5.7% higher than computed using RPA at MR of 3.8. An explanation for this deviation is not readily available but turns out to be not critical for the accurate trajectory simulations of reference 6.

Table 3: SpaceX Raptor 2 RVAC engine (vacuum variant) calculated technical data at 30 MPa and MR= 3.4

mass flow	kg/s	655.77
sea level thrust	kN	1943.05
vacuum thrust	kN	2369.64
sea level spec. impulse	s	302.14
vacuum spec. impulse	s	368.48

Despite some deviations of the calculated thrust levels compared to published information on Raptor, the DLR performance estimation is in overall good agreement.

1.1.3 Expected payload performance of Starship V1 and V2

Validation of the launcher models of DLR has been achieved by recalculation of the performed Integrated flight tests (IFT) from Boca Chica in the US. Reference 6 describes the Starship's and Super Heavy's ascent, return and reentry trajectories for IFT-2, IFT-3 and IFT-4 and compares with simulation data. The models described and calibrated with telemetry data from IFT#2-4 are used to extrapolate the Starship's payload to orbit performance for fully reusable operations [6].

Two configurations are investigated: the current Starship V1 and the enlarged Starship V2 proposed for the future. The V2 announced in April 2024 [10] is supposed to represent the vehicle in an early operational phase after its ongoing

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test phase. The key differences to the current test version are a larger propellant loading, a slight increase in length, and the use of the more capable Raptor 3 engines. The configurations' key parameters are listed in [6]. The maximum possible payload is estimated for the direct ascent into a 250 km x 300 km orbit with an inclination of 26°, flying directly eastward from the SpaceX's Boca Chica facility. These orbit parameters are close to the reference mission considered for European heavy lift-launchers (see section 2.1). Any dogleg maneuvers in order to avoid overflying populated areas during ascent are disregarded, having minor impact on performance in this case.

In the simulations, the fully reusable Starship V2 configuration achieves a gross payload to LEO of 115 t (Table 4). This almost doubles the payload capability of the simulated Starship V1 configuration and reaches the announced 100+ t. With this immense capacity, the configuration would surpass the largest recent launcher test-flown in 2022, the expendable Block 1 Space Launch System (SLS). If an engine mass of Raptor 3 as published by SpaceX [9] at 1720 kg is assumed, the gross payload would increase further to 125 t [6]. The deployable payload is of more practical interest. This is called here net payload and subtracts the payload attachment structure and considers for the Starship a mass contingency for sufficiently large payload bay doors. Such a design and hence any precise mass data are unknown. The assumption of 14 t on V1 and 15 t on V2 are reasonable first guesses and still would allow impressive 45 t to 100 t net payload mass in a fully reusable space transportation system.

Table 4: Payload and launcher mass of the V1 and V2 Starship &SuperHeavy in DLR modeling

	Starship & SuperHeavy **V1**	Starship & SuperHeavy **V2**
Propellant mass	4500 t	5150 t
Estimated Total Dry Mass	429 t	445 t
Lift-off mass (without payload)	4931 t	5596 t
Gross payload LEO	59 t	115 t
Net estimated payload LEO	**45 t**	**100 t**

2 European Heavy-Lift Launcher Options

Large space infrastructures as well as deep space missions could also require in Europe significantly more performant space transportation in the foreseeable future compared to what is existing today. ESA is starting to define and evaluate a "hub and spoke" space logistics network to reach the final orbits (e.g. constellations phasing, exploration missions…) and provide transportation support for in-orbit servicing (see e.g. [11]).

Together with exploration ambitions to the Moon or interplanetary, this infrastructure will require efficient means of transportation from Earth to LEO. The technical concepts of future European heavy-lift orbital launch capabilities are linked to the following development targets: short-term up to 2030, medium-term: after 2035 up to 2040 and longer term around 2050. This classification will later also be used in the discussion of a potential development roadmap.

2.1 Reference mission

All configurations in this section are assuming similar key mission requirements:

- 250 km x 300 km with an inclination of 25°
- Launch site: CSG, Kourou, French Guiana

The orbit represents a suitable staging orbit for a translunar trajectory but could also be representative for large LEO-satellite constellations.

The vehicles should be capable of performing secondary missions which were not investigated in the context of this paper. All expendable upper stages are to be actively deorbited at the end of their Earth orbital missions to reduce the buildup of additional space debris. A contingency of fuel mass is reserved for this final part of the mission.

2.2 Launcher main propulsion

Appropriate options of the main liquid stages' propulsion are all based on cryogenic fuels: either liquid methane or liquid hydrogen. Baseline here is a DLR-performed systematic assessment of future European engine concepts suitable for RLV-applications [30]. This investigation considers hydrocarbon propellants as well as hydrogen under similar conditions and preliminary high-thrust engine designs of 2200 kN vacuum thrust level in gas-generator and staged-combustion cycle. By purpose, the assumed main combustion chamber pressures are less ambitious than those of SpaceX Raptor (see section 1.1.2).

2.2.1 Open Gas generator cycle engine PROMETHEUS

PROMETHEUS is the precursor of a new European large-scale (100-tons class) liquid rocket engine using methane as fuel. Currently, the precursor of PROMETHEUS is under development (Figure 1 showing the prototype without nozzle

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extension). The calculated data in Table 5 have been generated by DLR to make realistic performance of a full-scale engine available for the launcher system design. The intention of this paper is *not* to provide an accurate prediction of the future PROMETHEUS for which technical characteristics are not yet all frozen.

Table 5: Calculated technical data of gas generator Methane engine as assumed for potential Ariane 6 liquid boosters

Mixture ratio [-]	2.67
Chamber pressure [MPa]	11.77
Mass flow per engine [kg/s]	421
Expansion ratio [-]	16.4
Specific impulse in vacuum [s]	316
Specific impulse at sea level [s]	288
Thrust in vacuum [kN]	1305
Thrust at sea level [kN]	1200

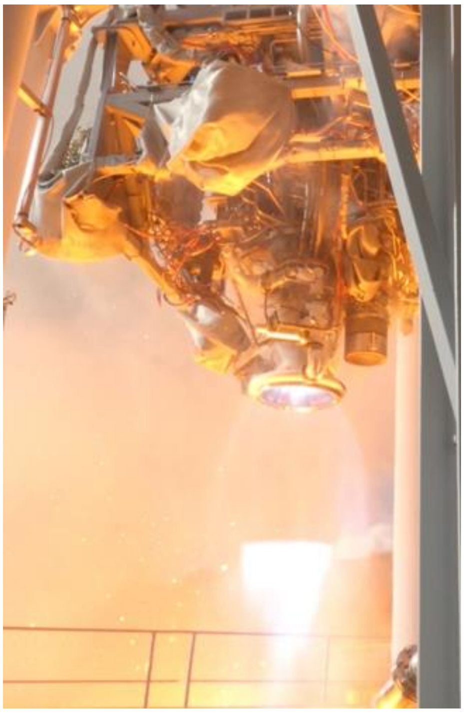

Figure 1: PROMETHEUS engine during early hot fire testing in Vernon 2023 [12], Courtesy: ArianeGroup

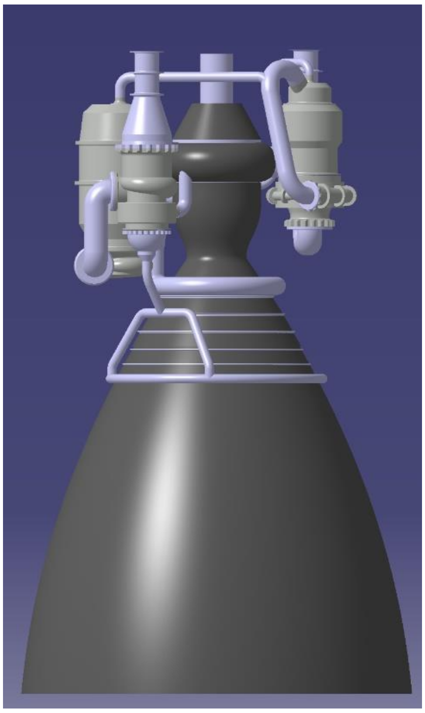

Figure 2: CAD-drawing of SLME V7 with nozzle expansion ratio 59

2.2.2 Staged combustion cycle engine SLME

A Full-Flow Staged Combustion Cycle with a fuel-rich preburner gas turbine driving the LH2-pump and an oxidizer-rich preburner gas turbine driving the LOX-pump is the preferred design solution for the SpaceLiner Main Engine (SLME). It is interesting to note that the ambitious full-flow cycle is currently developed by SpaceX for its Starship&-SuperHeavy with the Raptor-engine [4]. The Swiss company SoftInway and DLR jointly completed in 2024 a de-risk study for ESA on the SLME-type rocket engine.

The expansion ratios of the SpaceLiner booster and passenger stage / orbiter SLME engines are adapted to their respective optimums; while the mass flow, turbo-machinery, and combustion chamber are assumed to remain identical in the baseline configuration

The SpaceLiner 7 has the requirement of vacuum thrust up to 2350 kN and sea-level thrust of 2100 kN for the booster engine and 2400 kN, 2000 kN respectively for the passenger stage. All these values are given at a mixture ratio of 6.5 with a nominal operational MR-range requirement from 6.5 to 5.5. The full pre-defined operational domain of the SLME is shown in [23] including extreme operating points. Table 6 gives an overview about major SLME engine operation data for the nominal MR-range as obtained by cycle analyses [23]. Performance data are presented for two different nozzle expansion ratios: 33 and 59.

The lay-out of the SLME V7 is relatively conventional, like SSME, and a more advanced version with ox-rich powerhead mounted in-line with the main combustion chamber as on Raptor is under investigation for the SLME. The engine

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masses of V7 are estimated at 3500 kg with the large nozzle for the upper stage and at 3218 kg for the booster stage [23].

Table 6: SpaceLiner Main Engine (SLME) technical data from numerical cycle analysis [23]

Operation point	**O1**	**O1**	**O2**	**O2**	**O3**	**O3**
Mixture ratio [-]	6		6.5		5.5	
Chamber pressure [MPa]	16		16.95		15.1	
Mass flow rate in MCC [kg/s]	513.5		555		477.65	
Expansion ratio [-]	**33**	**59**	**33**	**59**	**33**	**59**
Specific impulse in vacuum [s]	436.9	448.95	433.39	445.97	439	450.56
Specific impulse at sea level [s]	385.9	357.77	386.13	361.5	384.2	352.6
Thrust in vacuum per engine [kN]	2200	2260.68	2358.8	2427.28	2056.7	2110.49
Thrust at sea level per engine [kN]	1943	1801.55	2101.6	1967.32	1800	1651.56

2.2.3 Staged combustion cycle engine derivatives

The calculated characteristics of the staged-combustion engines used by DLR for the fully reusable PROTEIN-related configurations are presented in Table 7 and Table 8 and have been derived of SLME and the engine definitions in [30].

Table 7: PROTEIN sea-level staged-combustion engines calculated technical data

Fuel type	LCH4	LH2
Nozzle area ratio [-]	36	
Chamber pressure [MPa]	20	
Mass flow engine [kg/s]	642.6	516.7
Thrust at sea level engine [kN]	1976	1972
Thrust in vacuum engine [kN]	2200	
Specific impulse at sea level [s]	313.4	389
Specific impulse in vacuum [s]	349	434

Table 8: PROTEIN vacuum staged-combustion engines calculated technical data

Fuel type	LCH4	LH2
Nozzle area ratio [-]	120	
Chamber pressure [MPa]	16	
Mass flow engine [kg/s]	610.2	488.6
Thrust in vacuum engine [kN]	2200	
Specific impulse in vacuum [s]	367.5	459

2.3 Expendable: Ariane 6 derived heavy launcher

Europe's Ariane 6 has been under development since 2014 in two configurations: A62 with two solid strap-on boosters and A64 with four solid strap-on boosters. Ariane 6 has performed its inaugural flight on 9th July 2024 from Kourou to medium inclined LEO, releasing several CubeSats [13]. This launcher has a central core consisting of the Lower Liquid Propulsion Module (LLPM) equipped with Vulcain 2.1 engine and providing space for about 154 t LOX/LH2-propellants. On top of this stage is the upper stage (Upper Liquid Propulsion Module or ULPM) which is propelled by the re-ignitable Vinci engine [14]. On the side of the LLPM solid boosters provide additional acceleration to the launcher. These P120C boosters house about 142 t solid propellant. The maiden launch was performed with the Ariane 62 version, using two solid boosters while the more powerful Ariane 64 using four P120C is expected to have its debut in the near future.

For all the Ariane 6-derived configurations of this section a fairing mass of 2.6 t has been assumed.

2.3.1 Ariane 6 "Block2"

A first upgrade of the Ariane 6 called "Block2" is already under development and related work focuses on increasing the performance by increasing size and loading of the solid booster, now named P160. Compared to the P120C, 14 t additional solid propellants are added to each booster and the casing is increased in length by 1 m while maintaining the same outer diameter. This improved performance version of Ariane 6 is of particular interest for the heavy lift role. The thrust profile of the P160 used in the ascent flight optimization of DLR follows the law as presented in [15].

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This leads to a calculated payload of approximately 22.5 t into the reference LEO-mission, a gain of 3 t compared to the Block1 Ariane 64 version in that orbit. Assuming the feasibility of an underloading of 7 t fuel on the ULPM, a maximum gain of about one additional metric ton seems possible.

2.3.2 Ariane 6 Evolution with C130

Another option of future Ariane 6 evolvement (not yet decided) could be the replacement of the P120C/P160 SRM with potential new liquid booster "strap-ons" using LOX/LCH4 or LNG with PROMETHEUS engine (see section 2.2.1). Such a launcher concept might operate the side boosters in expendable mode for high-performance missions (as investigated here) or in RLV modes DRL or RTLS in medium / low performance missions described in [14].

The architecture intends to keep the existing liquid propulsion main stages LLPM and the ULPM including the inter-stage structure untouched as far as possible. The P120C (or P160) side booster of the Ariane 6, however, are replaced by new liquid boosters. This requires the same axial position for the booster attachment points at the H150's intertank and aft-skirt structure as for the Ariane 6 (see figure in [14, 16]). The feasibility assessment presented in [14] targets the maximum LOX/LCH4 propellant loading under tight geometry constraints. The liquid booster diameter is slightly increased from 3.4 m of P120C to 3.6 m instead. Both propellants are stored aft of the forward attachment ring in an integral tank with common bulkhead. Three PROMETHEUS engines are placed in linear arrangement in the booster's base. This design enables to put one of the engines in a center position simplifying the vertical landing with single operational engine in case of RLV-mode. The approximate nozzle exit diameter of 1.1 m is still compatible with this placement and the booster diameter [14].

The preliminary stage architecture shows that each booster can carry up to 130 t of propellant in total (C-130) and stage lift-off mass is estimated at less than 150 t. For the envisaged heavy LEO-mission, the C130 boosters are operated in expendable mode, that is without recovery in order to maximize the performance. This configuration would enable a performance of above 23.9 t into the considered orbit, 1.4 t more than the "Block 2" variant. This is possible despite the significantly lower GLOW due to the significantly better specific impulse of the LCH4-engines compared to solid propellants.

2.3.3 Ariane 6 Derived with closed-cycle engine on LLPM

Another potential A6-evolution could be the replacement of the Vulcain 2.1 engine on the central core by a high-thrust, better performing liquid engine. Straight forward is the installation of a LOX-LH2-staged combustion engine as the main propulsion system on LLPM. A suitable engine under investigation by DLR is the SLME (see section 2.2.2) which is required to be operated at mixture ratios between 5.5 and 6.5 [23], see also Table 6.

Operating the SLME at a mixture ratio of about 6 allows maintaining the existing tank setup of LLPM since the Vulcain 2.1 operates at MR of nearly 6 as well. Feed system and thrust structure will have to be adapted on A6 to the higher massflow of SLME. This impact on stage dry mass was not investigated here, but the increased mass of the SLME was considered.

Implementing the SLME on the Ariane 64 "Block 2" would boost the performance to a staggering 27.4 t that is a gain of 4.8 metric tons. A reduced mixture ratio of 5.5 would offer slightly better specific impulse but would imply a smaller LLPM propellant loading and a moderate stage re-design. The obtained performance of a lower mixture ratio is marginal at best, not justifying the re-design effort of the tank system.

A delayed ignition in flight of the SLME by approximately 51 s after lift-off in an altitude of 9.6 km allows maximizing the performance remarkably, carrying the payload to nearly 29.9 t. The ground impact point of an the LLPM would be shifted slightly to the East but still very far from any coastal areas. Obviously, the boosters need to be capable to assume full attitude control during this initial ascent with an inactive core propulsion and it might proof necessary to re-assess the structural strength of the booster attachments and possibly of the core stage itself. Due to the significant thrust increase by replacing the Vulcain 2.1 engine by the SLME, the loads on the structure increase as well compared to the "Block 2", both in terms of dynamic pressure (~20%) and axial load factor (~10%). This might lead to the need to strengthen the involved structure.

2.3.4 Ariane 6 Derived Synthesis

Figure 3 gives a depiction of the optimized ascent trajectories for the Ariane 6 evolution options in the heavy-lift LEO-mission. Tracks show some coherence with the achieved payload performance (Table 9). While the A64 "Block 2" and A6 Evolution with C130 pursue a very similar trajectory with the latter being slightly more performant during booster ascent, the trajectories using the SLME show a significant increase in acceleration, most noticeably during the flight of the LLPM after booster separation.

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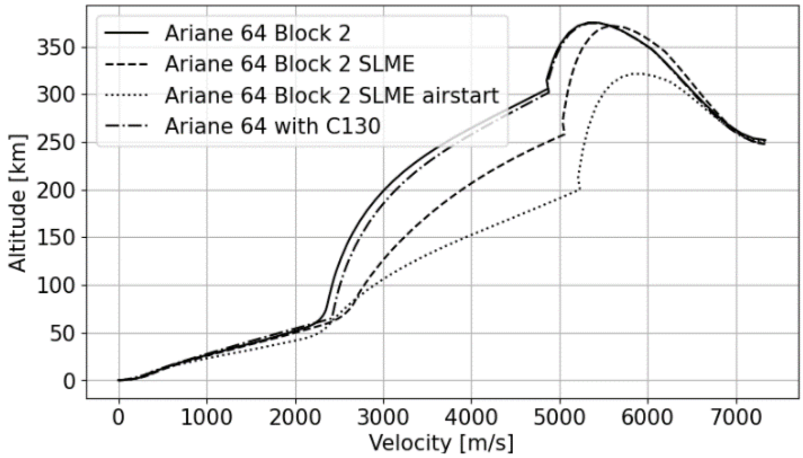

Figure 3: Ascent trajectories for Ariane 6 Evolution options in reference LEO

The version with altitude-ignition of the SLME is notably different from the other trajectories and its trajectory shows a more evenly and consistent altitude-velocity increase during the LLPM flight whereas the other variants perform a steeper ascent but then achieve lower MECO-velocity. The higher the initial velocity of ULPM, the lower its Δ-v-requirement and hence the more payload mass can be lifted.

Table 9: Payload and total mass of Ariane 64 Block 2 and options for Ariane 6 Evolution

	Block 2	C130 booster	SLME on LLPM	SLME-LLPM, air start
Lift-off mass (without payload)	916 t	806 t	917.6 t	917.4 t
Gross payload 250 km x 300 km x 25°	22574 kg	23983 kg	27421 kg	29909 kg

The gross payload mass in Table 9 includes also the mass of the payload adapter. The net performance hence might vary depending on the actual adapter mass to be foreseen and could be at least several hundred kg below. The small reduction of lift-off mass for the Ariane 6 Evolution with an air start of the LLPM is due to the fact that the solid boosters will consume slightly more fuel until the thrust is sufficient for release lacking the thrust of the central stage in this moment.

2.4 Partially reusable VTHL: RLVC-4

Investigations of semi-reusable heavy launchers with the internal project name RLV-C4 [14, 17, 18, 19] have been carried-out by systematic variation of design options on propellant choice or aerodynamic configuration. One concept with the SLME (see section 2.2.2) as the main engine has served as RLV-reference in the FALCon-project [24] and its architecture has some similarities to the SLB8V3 (section 2.5), however, with significantly reduced propellant loading (380 Mg) and only four SLME [17, 18]. An extensive cost estimation for launch vehicle families considering uncertain market scenarios revealed the RLV-C4 family reaching lowest recurring cost [20] of all investigated concepts. In contrast to the current article, the focus at that time was on the entire range of space transportation missions.

The RLV-C4 could form Europe's first step to reusable space transportation with a payload performance equivalent or even in excess of an expendable Ariane 6 evolvement as described in the previous section 2.3.3. The system studies at DLR's space launcher system analysis department SART have investigated not only one preferred type but different return and recovery modes, as well as different propellant and engine cycle options [14, 17]. Beyond the winged VTHL-concepts in focus of this section, similar VTVL options in architecture, size and payload performance have been studied as a potential alternative [14] and might be considered in future work. If smaller gas-generator type engines in the PROMETHEUS-class are implemented instead of the SLME, convergent designs have been found. However, in case of methane fuel the GLOW and hence the number of engines will dramatically grow [18].

Approaching or even exceeding the payload performance expected for Ariane 6 in GTO or Lunar exploration missions would require extremely tall launcher configurations in case of tandem-staged TSTO with reusable first stage. Therefore, for this class of RLV a parallel stage-arrangement is preferable: a winged stage is connected to an expendable upper segment with potentially various internal architectures. A 14 tons GTO-class with multiple payload capability can be achieved by a 3-stage architecture while still remaining at relatively compact size [14, 18].

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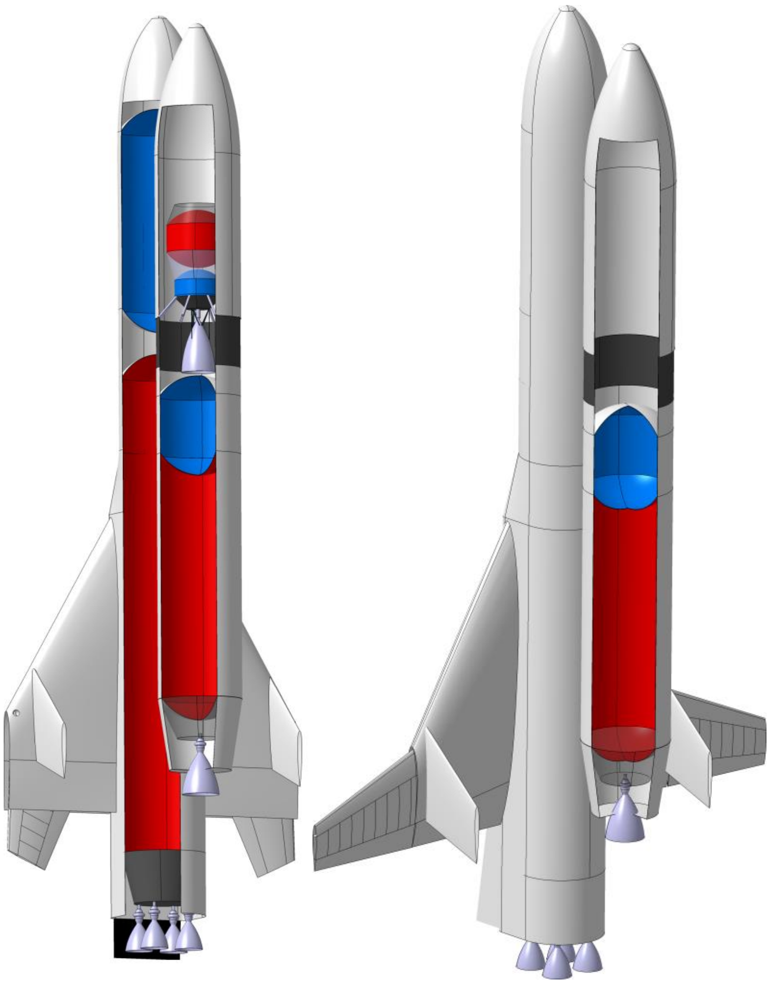

Figure 4: Launcher architecture sketches of RLVC-4-B configuration as 3STO (left), TSTO (right) [18]

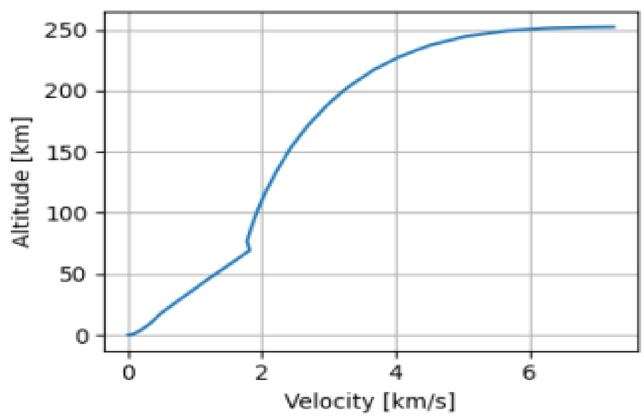

Figure 5: Calculated ascent trajectory of RLV-C4 in transfer orbit up to expendable second stage MECO

The TSTO-concept with large expendable 2nd stage (Figure 4 right) was initially defined as an H150, even more compact than the core stage of the classical Ariane 5G. With the heavy-lift LEO mission in mind, the expendable upper stage's propellant loading has been optimized keeping the single SLME untouched. The reusable RLVC-4 stage remains also unchanged.

A small increase in propellant loading to 160 Mg is delivering the optimum performance with roughly 27900 kg separated payload. The 2nd stage would grow slightly in length compared to what is shown in Figure 4. The disadvantage of bringing this stage into LEO is the requirement of its controlled deorbitation consuming a significant amount of fuel. Therefore, the interest of using a 3STO instead has also been studied for the same mission which allows the large cryogenic stage remaining suborbital and automatically splashing into the Pacific.

A hypothetical storable kick-stage has been defined for raising the orbit from second stage MECO of 30 km x 250 km (this initial ascent plotted in Figure 5) to 250 km x 300 km. A separated payload of around 29350 kg could be reached, an improvement of approximately 1445 kg compared to the TSTO.

If the final destination of the mission is similar to the reference LEO a fully cryogenic 3STO with H14 3rd stage (as visible in Figure 4 at left) is of limited interest being too heavy and too expensive. However, in case of more demanding missions the picture is changing and such a configuration could become highly attractive for e.g. translunar injection.

2.5 Partially reusable VTHL: RLVC-5 with SpaceLiner SLB8

A semi-reusable launcher based on the new SpaceLiner 8 booster design [21, 22] and a side-mounted large expendable upper stage has been defined under the designation RLVC-5 (Figure 6). The configuration's architecture is quite similar to the RLVC-4 TSTO (section 2.4) but a significantly larger winged RLV-stage with 10 SLME (see section 2.2.2) instead of merely 4. In comparison to the previous SLB7-3, the fuselage diameter is increased to 8.8 m. As a consequence, the stage length reduces to 79.1 m (without body flap).

The principal architecture of the expendable stage is even more similar to the RLVC-4 TSTO's second stage with LOX-LH2 stored in a common bulkhead tank and powered by a single SLME in large expansion ratio variant. Faring

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and hence stage diameter has been increased to 6.5 m. A huge 24.2 m long fairing that provides 700 m^3 of internal volume is assumed for the super-heavy lift transport with its mass conservatively estimated at 6400 kg.

At lift-off the ten engines on the RLV-booster stage SLB8 are ignited and accelerate to stage separation at high altitudes at the edge of the atmosphere. This maneuver could be relatively relaxed and could allow a certain delay in upper stage ignition if required or beneficial. While the SLB8 is kept in the configuration for its primary application of SpaceLiner, the expendable stage's propellant loading has been varied to find the maximum achievable payload mass.

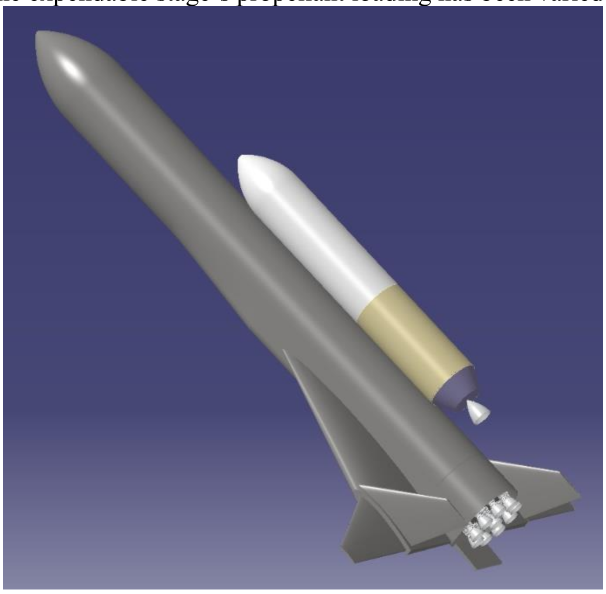

Figure 6: RLVC-5 as CAD geometry

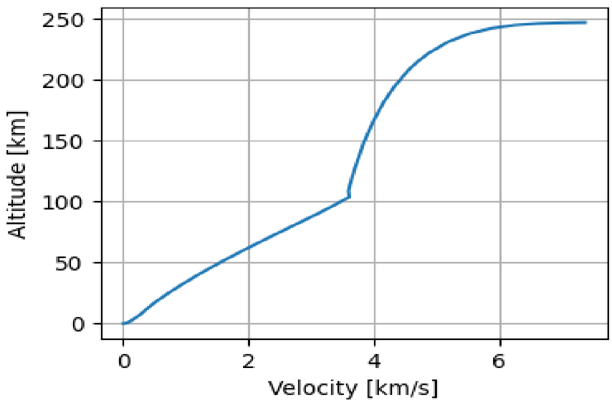

Figure 7: RLVC-5 ascent with expendable upper stage H160 in reference LEO

The investigations reveal that the maximum payload to the reference LEO is found at 80 t with an upper stage ascent propellant of approximately 160 t. Thus, the expendable part of this heavy launcher remains relatively compact in size (length 18.8 m without faring) as visible in Figure 6. Achieving maximum performance, would require some off-loading on the RLV-stage to realize adequate initial acceleration levels. Though this choice is not resulting in the optimum launcher for this particular application, the approach makes sense if the SLB8 designed for the SpaceLiner missions (e.g. [21]) is used for secondary tasks and thus demonstrates its operational robustness.

In Figure 7 the RLVC-5's direct injection into the 250 km x 300 km reference orbit is shown. RLV stage separation occurs at higher altitude but similar velocities as will be required for the fully reusable missions.

Return to the launch site of the SLB is traditionally assumed to make use of the patented "in-air-capturing"-method which likely provides the best performance [24]. Full simulations of the SLB-recovery are still open and should be performed in the future for the SLB8-configuration. The study for the next SpaceLiner 8 booster design is ongoing, however, a consolidated configuration is not yet defined. Despite some advantages, more analyses are needed to define the SpaceLiner 8 booster stage. Nevertheless, the SLB8V3 as summarized above, serves in this paper as the large, reusable booster stage of the RLVC-5 future European heavy-lift launcher.

2.6 Reusable VTHL TSTO SpaceLiner

The SpaceLiner is defined as a fully reusable space transportation system to LEO with payload performance in the A6-class. The parallel arrangement of the two SpaceLiner stages of variant 7, the reusable booster and the orbiter or passenger stage, at lift-off and its main dimensions are presented in reference 21. The version 7 is powered by 9 + 2 SLME (see section 2.2.2) at lift-off

The SpaceLiner7 passenger stage's internal design has been adapted for its secondary role as an unmanned satellite launcher. The passenger cabin is not needed for this variant and is instead replaced by a large internal payload bay [25] as shown in Figure 8. Key geometrical constraints and requirements are set that the SpaceLiner 7 passenger stage's outer mold line and aerodynamic configuration including all flaps should be kept unchanged. The internal arrangement of the vehicle could be adapted; however, maximum commonality of internal components (e.g. structure, tanks, gear position, propulsion and feed system) to the passenger version is preferred because of cost reflections.

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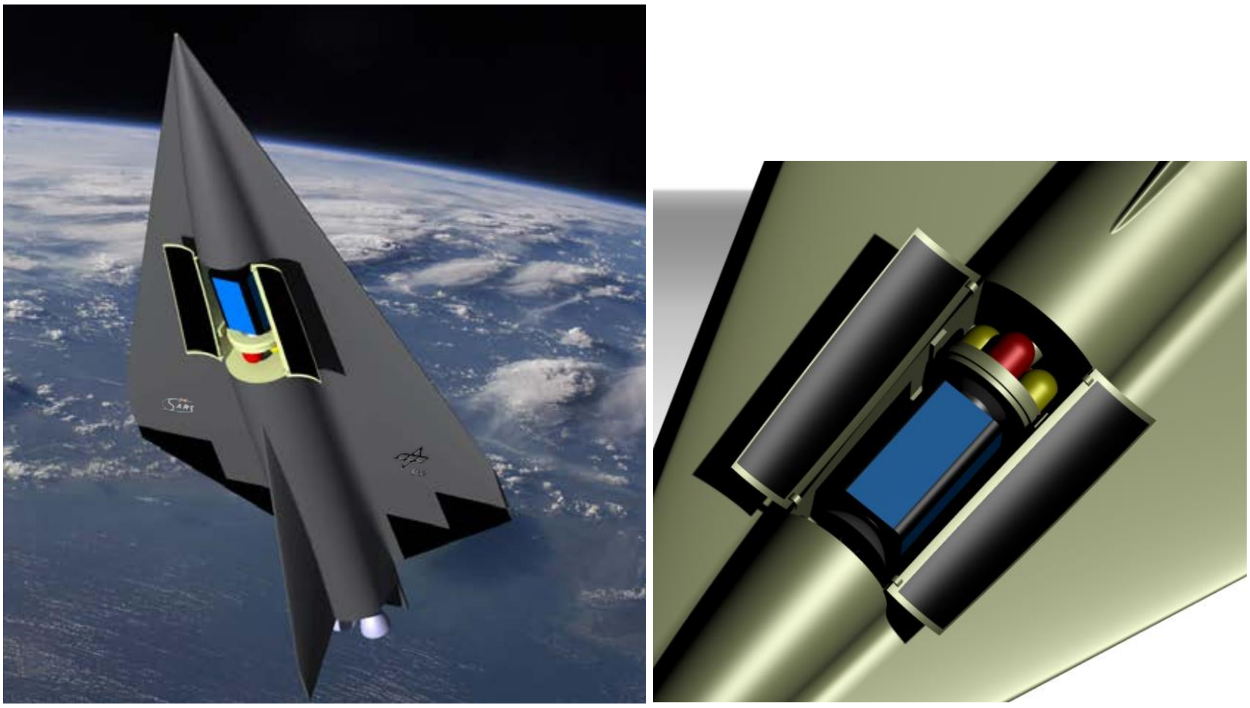

Figure 8: SpaceLiner 7 orbital stage (SLO) in renderings with open cargo bay and payload with kick-stage

Further, the payload bay should provide sufficient volume for the accommodation of a large satellite and – if required - its orbital transfer stage. For this purpose, the SLO's propellant loading has been reduced by 24 Mg to 190 Mg compared to SLP with a smaller LOX-tank to allow for a payload bay length of 12.1 m and at least 4.75 m diameter [25]. These dimensions are close to the Space Shuttle (18.3 m x 5.18 m x 3.96 m) and should accommodate even super-heavy GTO satellites of more than 8 m in length and their respective storable upper stage [25]. Large doors open on the upper side to enable easy and fast release of the satellite payload in orbit.

The SpaceLiner 7-3's GLOW as TSTO without payload is at 1783 Mg (Table 10) and reaches including the payload between 1800 and 1810 Mg. These values are below those of the partially reusable Space Shuttle STS of more than 2000 Mg with at the same time higher delivered payload by the SpaceLiner. This better launch efficiency results out of the fully cryogenic system and the SLO being unmanned, saving the crew cabin and life-support systems of the Space Shuttle.

Launch of the SpaceLiner 7 TSTO orbital launcher has been simulated in the past from the Kourou space center for various missions. In case of satellites transported to GTO, the injection of SLO occurs into a low 30 km × 250 km transfer orbit allowing the reusable orbiter stage becoming a once-around-Earth-vehicle capable of reaching its own launch site after a single circle around the planet. Subsequently, an orbital transfer is necessary from LEO to GTO using an expendable upper stage with storable propellants. Reference 21 shows the Mach-altitude-profile of the two reusable stages for the GTO-mission ascent.

The SpaceLiner 7 TSTO has been newly calculated for the reference LEO-Mission using a similar bi-boost strategy of the SLO as previously applied to the ISS-mission [25]. The initial ascent goes into a 70 km x 300 km LEO before the perigee is to be raised by second SLME burn to 250 km. Figure 9 presents the initial phase of the orbital ascent profile. Almost 20 tons separated payload could be delivered if the reusable upper stage itself is orbited (Table 10). The relatively large dry weight of the SLO makes it attractive even in case of the LEO-mission to consider keeping the orbiter in suborbital conditions and attaching a smaller expendable kick-stage for in-orbit injection ($\Delta v \approx 66$ m/s) of the payload. As a consequence, the achievable payload mass increases to more than 24 tons (Table 10) and overall complexity is reduced; e.g. an active deorbiting is not needed.

Table 10: Mass data of SpaceLiner 7-3 TSTO fully reusable launch configuration

GLOW without payload [Mg]	1783
Payload SLO 250 km x 300 km, 25° [Mg]	19.85
Payload by kick stage 250 km x 300 km, 25° [Mg]	24.25

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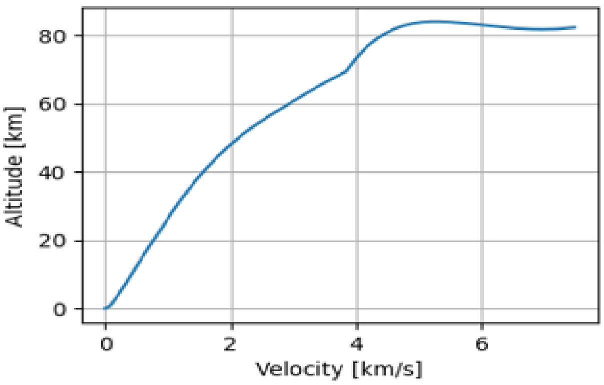

Figure 9: Calculated ascent trajectory of SpaceLiner 7-3 TSTO in LEO mission plotted up to SLO's 1st MECO

Figure 10: Final part of SLO descent trajectory ground track for LEO mission

The SpaceLiner Orbiter reentry has been simulated for both injection options. The aerodynamic trimming of the satellite transport stage with the SLP's trailing edge flaps and its bodyflap has been preliminarily checked in numerical simulation under hypersonic flow conditions of atmospheric reentry and is found feasible within the constraints of the 7-3 lay-out [25]. In case of SLO injection one full orbit is to be performed before deorbiting. Reaching the destination CSG in Kourou is without problem for the orbiter, either from orbit or simpler once-around mission due to its still very good hypersonic L/D. The vehicle crosses Central America at high altitude and turns to the South over the Caribbean Sea reaching CSG from the Atlantic (Figure 10). Note that the red color indicates where the sonic boom may reach the ground and that it is entirely above the sea. The maximum heatloads remain slightly lower than for the reference SL7-passenger concept because of a different AoA-profile and lower vehicle mass. The preliminary assumption of a common TPS with the passenger stage is once again confirmed by the new reentry simulations for LEO.

Although, the SLO TSTO-launcher has attractive launch costs for many missions [25], this fully reusable vehicle to LEO has not the capability of challenging the Starship&SuperHeavy concept.

2.7 Fully reusable VTVL: PROTEIN-related study

The ESA-initiated PROTEIN-study called for the creation of a concept capable of carrying up to 10000 tons of LEO-cargo per year using a new European Heavy Lift Launcher (EHLL) not exceeding the recurring payload cost target of 280 €/kg [28].

DLR joined Rocket Factory Augsburg (RFA) in PROTEIN for the definition of a fully reusable launch vehicle optimized for missions to LEO coupled with in-space transportation to final destinations [29]. However, this section *does not* summarize results out of the ESA-funded PROTEIN-study but additional configurations investigated later by DLR under similar but not necessarily identical boundary conditions.

In order to reach the ambitious recurring cost-objective, the EHLL is expected to be a fully reusable heavy TSTO-vehicle much larger than any other heavy launcher currently envisaged in Europe. The basic launcher architecture is inspired by the SpaceX Starship&SuperHeavy as VTVL TSTO with stages in tandem arrangement. Two variants have been considered and preliminarily sized in iterations: one based purely on methane fuel and the other storing methane in the 1st stage while switching to hydrogen fuel in the 2nd stage. The main propulsion system is based on closed cycle engines with chamber pressures up to 20 MPa as listed in section 2.2.3.

Figure 11 shows CAD-based sketches with LOX-tanks in blue, LH2-tank in red, and LCH4-tanks in green. Note, aerodynamic characteristics are reused from DLR's 2022 analyses of Starship [4], however, any dedicated assessment and flyability evaluation has not yet been performed for the PROTEIN-related types. Therefore, these devices are also not shown. It is interesting to see in Figure 11 that the "hybrid" configuration with both fuels, methane and hydrogen becomes significantly more compact than the pure LOX-LCH4-launcher. The hybrid configuration is about 98 m in length while the one with methane reaches about 108 m with increased diameter of the 1st stage.

Figure 12 gives an impression of possible engine arrangements in the base areas of the stages. As is already well-known from the SpaceX SuperHeavy, a purely methane-fueled reusable TSTO-launcher requires an extremely dense packaging of the engines on the first stage. In total 35 methane staged-combustion engines would be needed instead of merely 24 of the same engines on the first stage of the hybrid configuration. A skirt had to be added to accommodate all 35 engines in the base (compare Figure 11 at bottom right). The engines colored in red in Figure 12 have a dedicated role for vertical landing or hovering.

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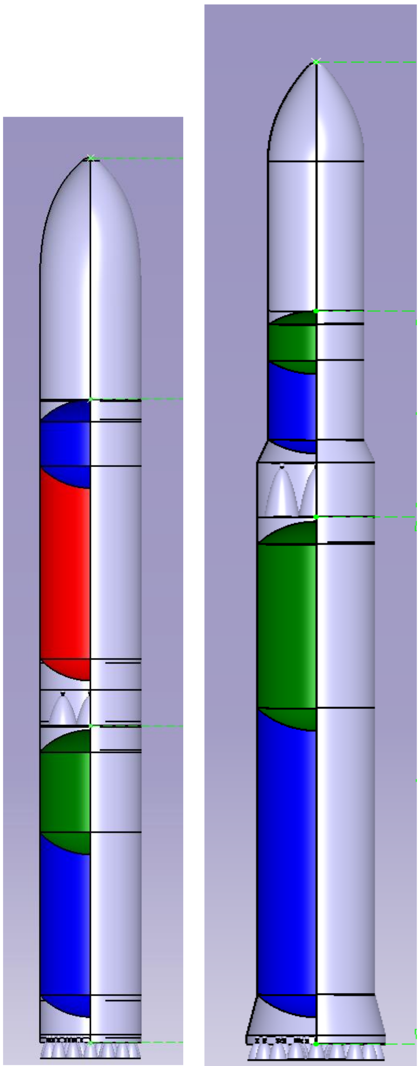

Figure 11: Architecture definition of EHLL "hybrid" (H_2 and CH_4) type (left) and only methane type (right) [31]

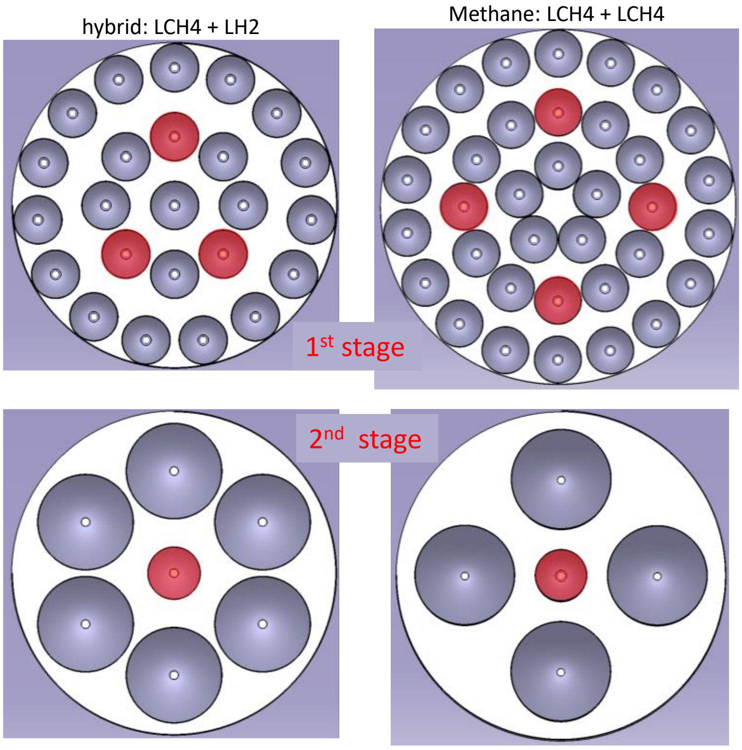

Figure 12: Engine arrangement of EHLL hybrid type (left) and methane type (right) [31]

All launcher configurations have been sized by trajectory optimizations for ascent and descent or return of the reusable stages. The launchers were all predesigned for high payload mass but not necessarily that all concepts reach the same values. Note in Figure 13 left the remarkably lower separation velocity of the hybrid launcher compared to the pure methane variant in case of first stage return to the launch site. It is advantageous to shift a larger portion of total Δ-v to the more efficient LOX-LH2-2nd stage.

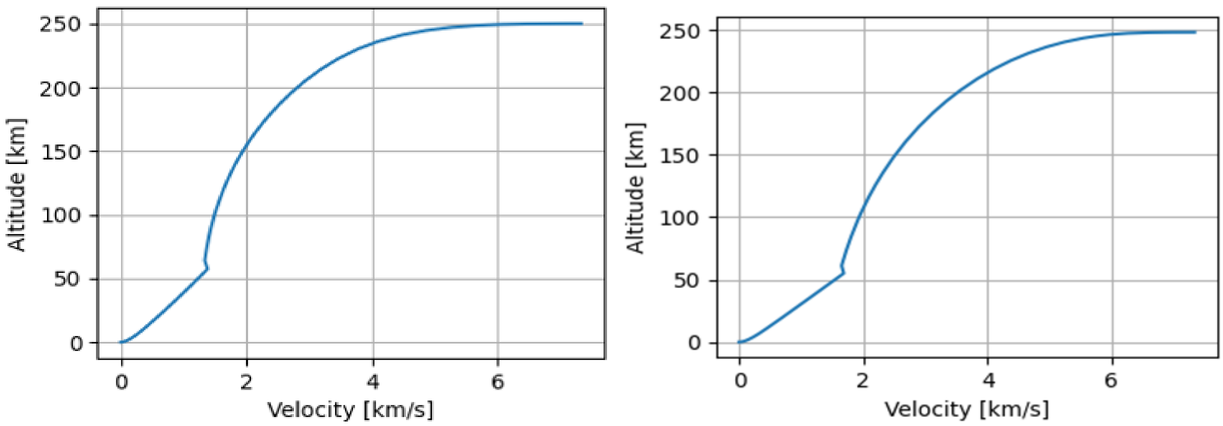

Figure 13: Calculated ascent in 250 km x 300 km reference LEO considering RTLS with "hybrid" (H_2 and CH_4) type (left) and only methane type (right)

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Despite the more compact layout, the calculated performance of the hybrid launcher with hydrogen in the upper stage has a significantly better payload performance (Table 11). This was already the case for the PROTEIN-related design mission of 450 km and inclination of 6° between +23% (DRL) and +115% in case of RTLS [16, 31]. The higher inclination of 25° in combination with the RTLS maneuver further augments the difference to almost +400%.

Table 11: PROTEIN-related EHLL calculated payload performances in initial and reference LEO for different 1st stage return modes

[tons]	DRL 450 km circular LEO at 6° [16]	RTLS 450 km circular LEO at 6° [16]	**RTLS reference LEO at 25°**
"hybrid" CH4+ H2	126.4 t	92.1 t	**72.7 t**
purely CH4	102.6 t	42.9 t	**14.65 t**

The relatively poor performance of a fully reusable methane-based super heavy TSTO is on the one hand due to the limited performance of this propellant combination. Further, the staging conditions turn-out to be not optimal and a shift of propellants from the first to the second stage would improve the situation. A systematic assessment by DLR with generic assumptions for fully reusable TSTO gives directions for future designs [32].

3 First indication of development roadmap

Having summarized the most important technical data of the European heavy-lift launcher options in the previous section, these must now be logically sorted in order to determine the possible implementation period. The potential roadmap in Figure 14 puts all the concepts on a time scale for the next 25 to 30 years. Obviously, not all of these large configurations should actually be realized but some are better understood as alternative options.

Within the next roughly 10 years only expendable heavy-lift launchers are to be expected. The A6 Evolution with potentially reusable liquid strap-on boosters [14] can serve the payload class above 20 t to LEO only in fully expendable mode. Partially reusable systems might be realized with the cryogenic RLVC-4 and -5 starting from the second half of the 2030s, potentially achieving after 2040 significant payload mass of up to 80 t. Following a sober assessment, a fully reusable TSTO bringing more than 20 tons to LEO is not to be expected before end of the 2040s. Any such vehicle in the class of 100 t comparable to Starship&SuperHeavy will realistically require at least another 25 years before becoming operational for Europe.

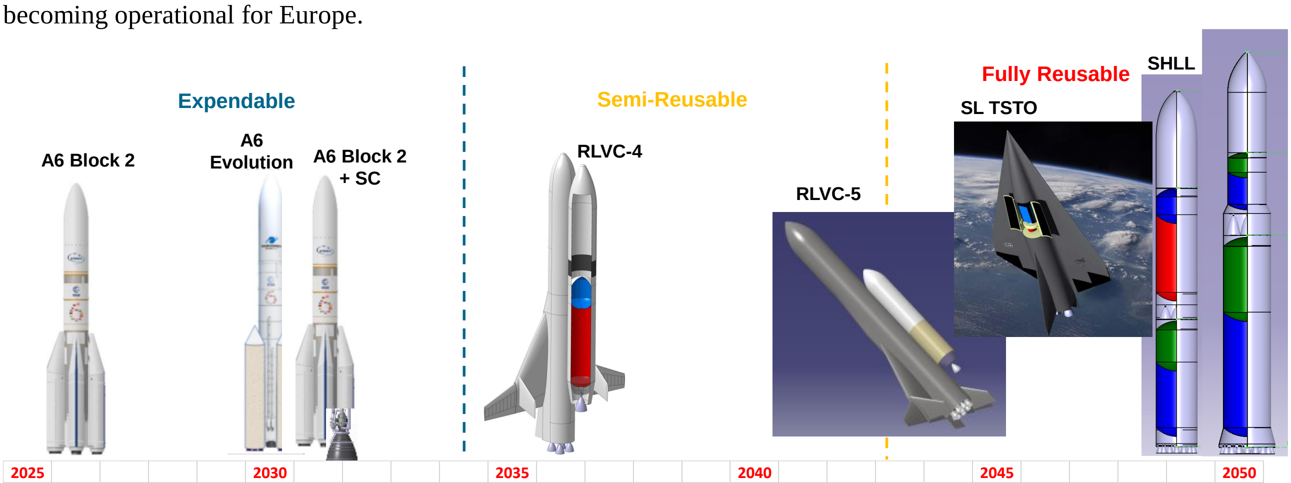

Figure 14: Potential European roadmap for heavy-lift launchers in the next 25 years

The answer to the question, which of the launchers presented in Figure 14 are to be realized, strongly depends on the overall operational scenario of European space transportation. If performance should approach 30 tons relatively soon, a high thrust closed-cycle engine to be integrated in the A6 core stage is attractive. An engine such as the SLME could be matured first in expendable operations before being attached to the reusable first stages. Even if the engine might seem oversized for its initial application, the elevated thrust-level in the 2200 kN class will pay off in all future heavy-lift launchers. In case a broad range of missions should be served and available development budgets might not excessively grow, a semi-reusable option as RLVC-5 carrying heavy-payloads and the same RLV-booster accelerating also the fully reusable upper stage of SL TSTO with missions of lower demand could turn-out to be most attractive.

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4 Conclusion

DLR has modeled the SpaceX fully reusable super heavy-lift launcher Starship&SuperHeavy based on publicly available data from the flight tests (IFT#2-4) and engineering analyses. Based on such consolidated models, the performance into 250 km x 300 km LEO has been estimated. An impressive 45 t to 100 t net payload mass in a fully reusable space transportation system seems to be realistically available in the coming years.

The main part of the paper is dedicated to the technical evaluation of European options in serving a roughly similar payload class approaching 50 Mg in single launch to LEO.

A launcher system analysis first looks into Ariane 6 evolution options and explores the technical limits based on the assumption of expendable stages. The goal of 50 tons can't be reached by any of the investigated evolutionary A6 concepts. Nevertheless, LEO-performance could be pushed up to values between 22.5 t and more than 29 t. The maximum gain is achieved when the LLPM propulsion of Vulcain 2.1 is replaced by a high-thrust staged-combustion engine. It is acknowledged that results on modified existing launchers are to be taken with care because potential constraints on controllability and structure are difficult to assess without deep knowledge of the existing design.

A significantly better performance perspective can be achieved through a completely new architecture based on building blocks that already exist or are under development. In case of the new architecture launchers, all first stages are designed reusable and exclusively liquid cryogenic propellants are chosen. The RLVC-4-concept of DLR might reach similar performance as the A6-derived variants, however, with a major reusable part. While the fully reusable TSTO SpaceLiner orbiter with internal payload bay based on previous studies is limited to the payload range below 25 tons, using the latest variant of the reusable SpaceLiner booster in combination with an expendable cryogenic upper stage would allow Europe entering the super-heavy class with 80 t payload.

Fully reusable configurations have been addressed in the small PROTEIN-study of ESA for which some complementary DLR-concepts are presented. Completely new designs of considerable dimensions will be required if payload mass to LEO should approach or even exceed 100 tons. The combination of methane lower stage and hydrogen upper stage is of great interest as the payload delivered would be significantly superior even with a more compact lay-out compared to the pure methane concept.

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